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1.0 INTRODUCTION AND RESEARCH OBJECTIVES

This document constitutes the final technical report on a multi-investigator research program carried out at the MIT Gas Turbine Laboratory on unsteady and three-dimensional flow phenomena in turbomachines. The objectives of the research program are:

- (1) To determine the aerodynamic performance penalties due to film cooling on a fully-scaled turbine stage through quantitative delineation of unsteady, 3-D flow and heat transfer processes.
- (2) To examine, experimentally and computationally, novel techniques for suction enhancement of the pressure ratio of high performance compressor stages and effects of suction on compressor efficiency, and to quantify the benefits to engine systems of the use of fluid extraction in the compressor.
- (3) To develop boundaries for use in flutter clearance which span full relevant parameter space so that adequate margins can be established to ensure absence of aeromechanical problems at all operating points.
- (4) To quantify the effects of tip clearance flows on multistage compressor performance as measured in terms of efficiency, pressure rise capability and operability.

A common theme that threaded through these research areas was three-dimensional and/or unsteady flow phenomena, and the enhancement of overall performance of turbomachines through flow management/design. In other words, it is our view that the primary challenges to achieving increased overall performance of turbomachines are linked to local phenomena which are inherently three-dimensional and/or unsteady.

1.1 Research Abstracts

Fully-scaled turbine stage measurements with advanced experimental diagnostics and measurements on two sets of experiments, cooled vs uncooled, have been completed. Results from these tests provided first-of-a-kind data for quantifying the effects of coolant interactions with main flow on turbine performance. Uncooled turbine test data elucidated the variations of isentropic efficiency with pressure ratio across the low pressure turbine (LPT) for two tip clearances. Specifically, there is a 2.5 point deterioration in efficiency at design and more at off-design when the tip clearance is increased from 1.5% to 3.0% (of blade height). This trend is also evident in the result for the variations of isentropic efficiency with reduced speed for two tip clearances.

The performance of compressors can be enhanced by the judicious removal of the viscous boundary layer fluid from the flow path. The effort combined computational design approaches with an experimental feasibility demonstration. The design studies indicated that removal of the

boundary layer fluid at critical locations could lead to approximately doubling the work done by a compressor stage for a given tip speed, while maintaining or even improving the adiabatic efficiency.

Modeling conducted on the compressor showed that a full definition of stall-flutter boundaries for high speed compressors and fans requires consideration of at least four dimensionless parameters: Mach number, reduced velocity, incidence, and the ratio of mechanical damping to relative air density. The modeling further defines the shape of the flutter boundaries in this four-space, in terms of a set of coefficients that must be determined empirically and by use of numerical models that embrace two of the parameters.

An analytical model has been developed to quantify and elucidate the effect of compressor asymmetric tip clearance, which induces unsteady flow phenomena on the scale of the compressor circumference (*i.e.* large compared to blade pitch) on changes to the steady-state performance (stability limit and peak efficiency) of multistage compressors. Experimental measurements taken on an engine company facility are in good agreement with the analytical model. Both analysis and experiment showed that the decrease in stability (stall margin) for an asymmetric clearance was closer to that obtained with axisymmetric flow at the maximum clearance than to that at the average clearance. Thus, the degradation in peak pressure and stable flow range is more nearly associated with the *maximum* level of the clearance asymmetry, rather than the average.

The role of, and the dynamic processes associated with, tip leakage/endwall flow in axial compressor rotating stall inception were examined based on multi-blade passage computations and available experimental measurements. This situation involves flow features that are initially on the scale of a blade passage. However, the resulting instabilities are coupled to the annulus eigenmode structure and thus exhibit length scales much larger than the blade pitch.

2.0 NEW FINDINGS AND ACCOMPLISHMENTS

Fully-scaled cooled and uncooled turbine stage tests have been completed to provide first-of-a-kind data for quantifying the effects of coolant interactions with main flow on turbine performance. Uncooled turbine test data elucidated the variation of isentropic efficiency with pressure ratio across the low pressure turbine (LPT) for two tip clearances.

The feasibility of operating aspirated compressors was established. For effectiveness the aspiration and the blade shaping must be treated as elements of an integrated design.

The decrease in compressor stability (stall margin) for an asymmetric clearance was closer to that obtained with axisymmetric flow at the maximum clearance than to that at the average clearance. Thus, the degradation in peak pressure and stable flow range is more nearly associated with the *maximum* level of the clearance asymmetry, rather than the average.

3.0 TASK I: CHARACTERIZATION OF FULLY SCALED, UNSTEADY FILM-COOLED TURBINE BLADE HEAT TRANSFER AND AERODYNAMICS (A.H. Epstein, G.R. Guenette)

This research is part of an effort to study the aerodynamic performance penalties due to film cooling on a fully-scaled turbine stage. A short duration turbine test facility was used for this research. Short duration test facilities have been used extensively to study turbine heat transfer because of their low operating cost, and because they enable turbines to be tested under fully-scaled conditions. As part of this research initiative, it was necessary to develop and demonstrate a technique to accurately measure the performance of a turbine in a short duration facility. The technique was first demonstrated using an uncooled turbine in order to establish a baseline with which to compare the performance of the cooled turbine. The uncooled tests also gave the opportunity to address the issue of non-adiabatic effects on the turbine performance. The test turbine has a design pressure ratio of 2.0 with a loading coefficient of 2.0. For the cooled turbine, the cooling mass flow for the NGV, rotor, and casing cooling account for 10% of the stage exit mass flow.

This test required incorporating several new features into the MIT Blowdown Turbine Facility. The turbine torque and shaft speed were measured to determine the power produced by the turbine, and the ideal power was determined by measuring turbine mass flow, pressure ratio, and inlet total temperature. The turbine was tested over a range of operating conditions by varying its pressure ratio and corrected speed. The turbine was also tested over a range of turbine inlet temperatures in order to address the importance of non-adiabatic effects. Denton (1993) examined this problem and found that the effects of heat transfer on the work output of a non-adiabatic turbine can be expressed as:

$$W = W_{is} - \int (1 - \frac{T_5}{T}) \delta Q - m T_5 \delta S_{irr}$$

Figure 3.1 shows an estimate of the non-adiabatic correction as a function of turbine inlet temperature. The turbine inlet temperature for the uncooled tests was approximately 350 degrees Kelvin and the non-adiabatic correction is 0.25%.

Figure 3.2 shows the measured turbine performance over a range of corrected speeds. The measured results are compared with 3D Navier Stokes CFD predictions. The predictions and the measured results show a similar trend with corrected speed. Figure 3.3 shows the results of an experiment to determine the influence of tip gap on the turbine efficiency.

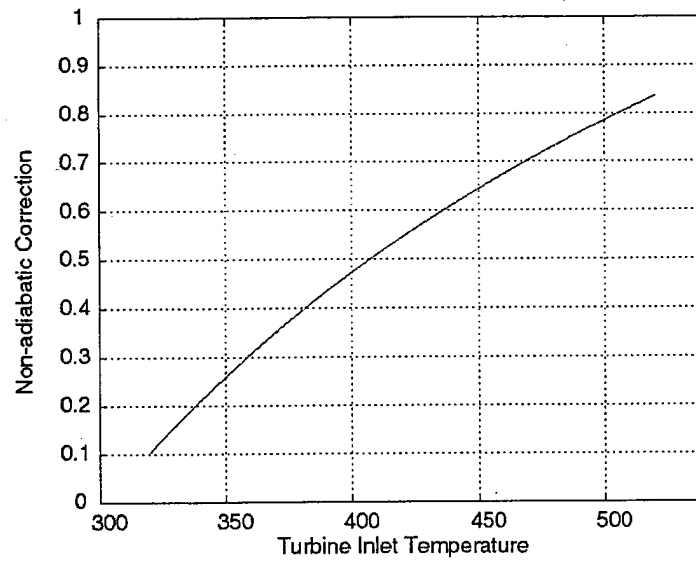


Figure 3.1: Non-adiabatic correction.

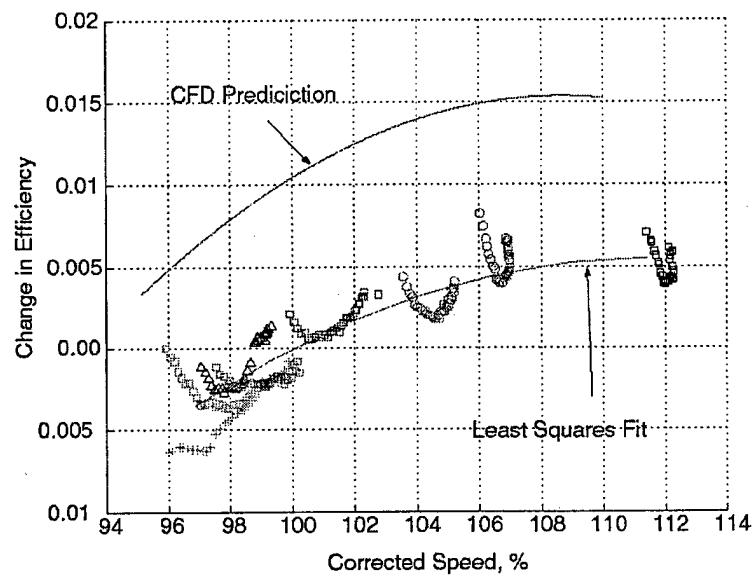


Figure 3.2: Efficiency change with corrected speed.

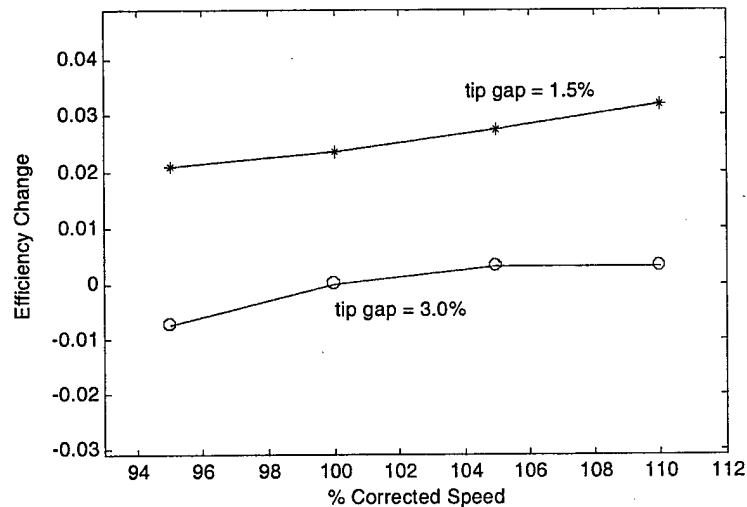


Figure 3.3: Efficiency change with tip gap.

3.1 Cooled Turbine Tests

The film cooled turbine was fabricated by incorporating cooling channels into the uncooled turbine nozzle guide vanes and rotor blades using electrical discharge machining. The film cooling holes were then machined by laser drilling holes from the airfoil surfaces to the cooling channels. The turbine was tested over the same range of operating conditions as the uncooled turbine. A range of coolant to mainstream mass flow and temperature ratios were tested. The data from the cooled tests is in the process being reduced. A combination of analytical and computational tools will be used to develop a model that predicts the impact of film cooling on the turbine performance.

3.2 References

Denton, J., 1993, "Loss Mechanisms in Turbomachines," *ASME Journal of Turbomachinery*, Vol. 115, pp. 621-656.

4.0 TASK II: FLOW CONTROL IN COMPRESSION SYSTEMS BY VISCOUS FLOW REMOVAL (J. L. Kerrebrock)

4.1 Flow Control in Compression Systems by Viscous Flow Removal (Sept. 1996 - Aug. 1998)

During the subject period, the AFOSR sponsored a research program at the MIT Gas Turbine Laboratory, exploring the use of boundary layer control by aspiration of compressors to improve the efficiency and the pressure ratio per stage. The program was conducted at the level of two or three graduate students and one faculty member part time. It combined computational design approaches with an experimental feasibility demonstration.

The computational effort consisted of the design by use of a unique quasi-three dimensional design approach of a family of compressor stages, ranging from a fan with pressure ratio of 1.6 to a compressor that was to deliver a pressure ratio of 3.5 at a tip speed of 1,500 ft/sec. The experimental component consisted of the incorporation of boundary layer removal on 5 blades of a 23-bladed transonic rotor that was available for test in the MIT Blowdown Compressor facility for modest cost.

Results of his effort may be summarized briefly as follows:

- a) The design studies indicated that removal of the boundary layer fluid at critical locations could lead to approximately doubling the work done by a compressor stage for a given tip speed, while maintaining or even improving the adiabatic efficiency. They also showed that to take the best advantage of the effects of boundary layer removal, the blade shapes must be carefully chosen. That is, the application of boundary layer control to existing blading is not very productive. Rather, the aspiration and the blade shaping must be treated as elements of an integrated design.
- b) The experiment showed that, even though aspiration on a fraction of the existing blades could not lead to a significant increase in the work of a compressor, the aspirated blades displayed higher loading capability than non-aspirated blades operating in exactly the same environment.

These results were reported in Kerrebrock *et al.* (1997).

4.1.1 Consequences of the AFOSR Program

With the encouraging results cited above, a more ambitious program was launched with DARPA support in July of 1998. It comprised several elements:

- 1) More detailed designs of two aspirated compressor stages, one with a pressure ratio of 1.6 at a low tip speed of 750 ft/sec and one with a pressure ratio of 3.5 at 1500 ft./sec. Both of these deliver approximately twice the work of non-aspirated stages at the same tip speeds.
- 2) Three-dimensional viscous analyses of the designs, which were fed back through the quasi-

three dimensional design process to improve the designs further.

3) Design, construction and test of the two stages.

The low pressure ratio stage was designed for test in the MIT Blowdown Compressor facility. It has been tested and has delivered its design performance, thus *establishing the feasibility of aspirated compressors*. The high pressure ratio stage is to be tested in NASA facilities. It is in construction and scheduled for test in the summer of 2000.

Kerrebrock *et al.* (1998). presents a status report on this program. Further references are listed in Kerrebrock *et al.* (1997) and (1998).

4.1.2 Summary

The initial support of AFOSR for a low-level academic research program has led to a multi-million dollar program involving academia, NASA and two engine manufacturers as well as the AFOSR, that promises to yield an entirely new design paradigm for compressors.

4.2 Compressor Flutter Clearance Methodology (Sept. 1998 - Aug. 1999)

4.2.1 Introduction

To assure reliability and safety of jet propulsion, designers must mitigate possible blade vibrations in the turbomachinery stages. The cyclical stresses associated with blade vibration can rapidly accrue to promote high cycle fatigue (HCF) failure, greatly diminishing blade life. The current work concerns the problem of stall flutter in compressors, a vibrational instability. In particular, the focus is upon developing a rational methodology towards "flutter clearance"; that is, towards ensuring that the boundary of the flutter instability lies outside the engine's operating envelope.

The current methodology of designating flutter regions on the performance map of pressure ratio in terms of corrected mass flow and corrected speed is useful, but incomplete. In particular, the corrected quantities do not scale out the dependence of inlet temperature and pressure on the flutter boundary, although they do scale out these dependencies upon performance (*i.e.* pressure ratio). Therefore, the boundary may "shift" (see Fig. 4.1) depending upon the inlet temperature and pressure. Failure to properly account for this dependency caused a major problem in the fan stage of the F100 Engine (Jeffers and Meece, 1975). Rig testing at sea level static conditions "cleared" the design, but subsequent flight testing showed that there was a flutter problem at sea level ram conditions. The shift in the flutter boundary is diagrammatically shown in Fig. 4.1. The redesign necessary to resolve this problem was extremely costly in terms of time, money, and engine efficiency. Subsequent studies further characterized the temperature and pressure dependency, but a good understanding of it has not yet been achieved. In the *AGARD Manual on Aeroelasticity*, Jeffers (1988) writes, "The adverse

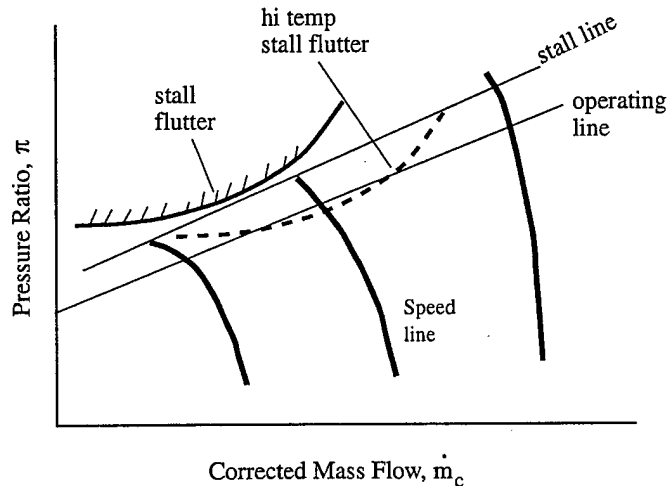


Figure 4.1: Performance map with stall flutter boundary. In the case of the F100 fan, an increase in inlet temperature (at sea level ram) caused the flutter boundary to encroach upon the operating line.

effect of increasing temperature on the stability of turbomachinery aerofoils has long been recognized but remains today one that is not fully understood..."

Towards the goal of constructing a rational flutter clearance methodology, we have taken the following steps:

- 1) A study to identify non-dimensional parameters for flutter stability.
- 2) A computational study of parameter dependence in an idealized case.
- 3) Preliminary analysis of fully-scaled engine flutter data.

4.2.2 Parameter Identification and Flutter Boundary Representation

To appropriately address the problem of flutter boundary migration, it is necessary to identify a sufficient set of non-dimensional operational parameters. We begin with a model of the rotor that considers a single mode of vibration for each blade, as schematically represented in Fig. 4.2a. Under a coordinate transformation (Crawley, 1988), the linearized system can be represented as a set of uncoupled single degree of freedom systems, each corresponding to a particular "interblade phase" angle between neighboring blades. This is depicted in Fig. 4.2b, in which a blade mass (for a given interblade phase) is supported by a mechanical spring and damper to the rotor. The least stable of all possible interblade phase motions is relevant for stability. The non-dimensional variables involved to describe the system stability are the following: Mach number, Ma , flow angle, α , reduced frequency, k , density ratio, ρ^* , and mechanical damping coefficient, g . That these five parameters are relevant to flutter stability is well known (*e.g.* Srinivasan, 1997).

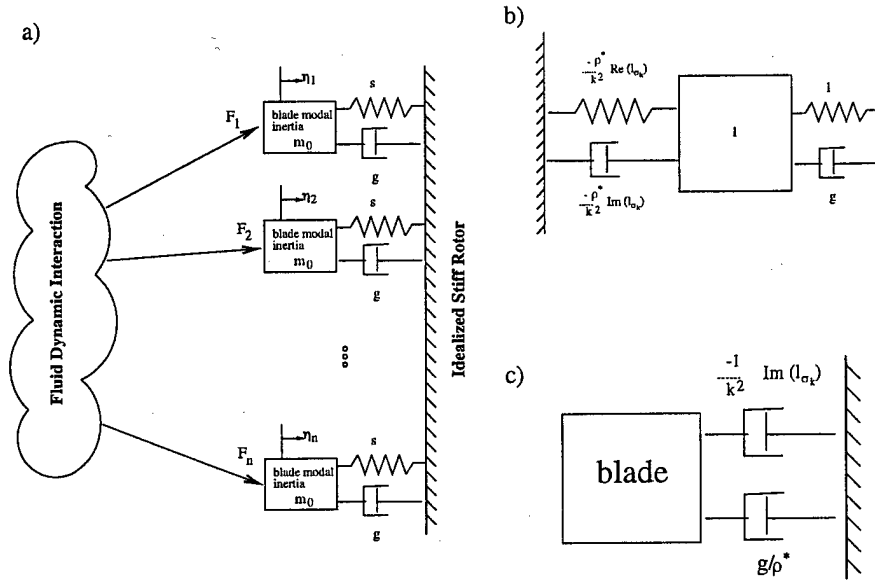


Figure 4.2: Schematic of model development: (a) shows the blades for a given vibrational mode. In (b) a transformation to interblade phase coordinates is made, which decouples the fluid effects. For purposes of linear stability, the system is equivalent to (c).

We can exploit the physical properties of the system to reduce the number of parameters to four. The fluid effects for harmonic motions are equivalent to a "fluid" spring and a "fluid" damper, corresponding to the in-phase (real) and out-of-phase (imaginary) components of the fluid force. The fluid forces depend upon a coefficient l_σ , which is a function of Mach number, flow angle, and reduced frequency, but is independent of density ratio. Also, the stiffness and inertia elements do not effect the system stability. Therefore, for stability purposes, the system is equivalent to the system shown in Fig. 4.2c. The stability criterion is that the total system damping should be positive, which can be expressed as

$$g/\rho^* > 1/k^2 \text{Im} (l_\sigma(Ma, \alpha, k))$$

where l_σ is the fluid force coefficient for interblade phase σ . The necessary number of parameters is, then, reduced to four: Mach number, Ma , flow angle, α , reduced frequency, k , and a combined mass-damping parameter, g/ρ^* . The combination of density ratio and mechanical damping into a single parameter for turbomachinery flutter is a novel insight obtained from this ground-up analysis. For inserted blades, the value of g/ρ^* is of order unity, in which case it may have a significant effect on flutter boundaries.

Alternate formulations of the four-parameter set may be useful as well. In particular, one promising concept we are currently developing is to replace the reduced frequency, k with a compressible reduced frequency, K^* such that $K^* = k Ma$. When used in conjunction with the

performance map, the resulting four parameter set becomes $(m_c, N_c, K^*, g/\rho^*)$. This formulation has the advantage that it decouples the location on the performance map (*i.e.* corrected mass flow and corrected speed) from the sensitivity to structural properties and inlet temperature and density. The definitions of K^* and g/ρ^* ,

$$K^* = \omega_c c / (\gamma R T_0) \quad \text{and} \quad g/\rho^* = g / (\rho_0 c^2 / m_0)$$

where c is the blade chord and γ and R are the ratio of specific heats and the gas constant, respectively, make it evident that K^* is dependent upon modal frequency, ω_0 and inlet temperature, T_0 , while g/ρ^* depends upon modal mass, m_0 , damping, g , and inlet density, ρ_0 .

Using this approach, it is then possible to describe the flutter stability of a machine in terms of a family of curves on the performance map in which each member has a specified K^* and g/ρ^* (see Fig. 4.3). The movement of the F100 fan flutter boundary, as shown in Fig. 4.1 can be interpreted as a shift in the relevant K^* for the curve. Also, by similarity, the "adverse effect of increasing temperature" described by Jeffers (1988) can be interpreted in terms of the well-known destabilizing effect of decreasing natural frequency.

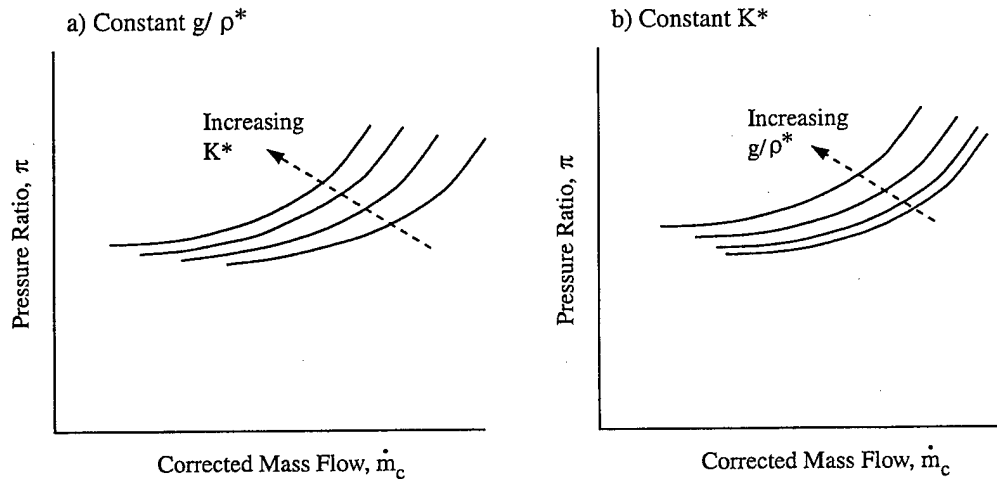


Figure 4.3: A family of flutter boundaries, showing trends with K^* and g/ρ^* on the performance map.

4.2.3 Exploration of Parametric Dependencies in an Idealized Case

Because of the limited quantitative fidelity of flutter prediction modeling, and the limits of the available data, it is useful to explore idealized, physically consistent models of flutter in developing approaches relevant to the actual case. We chose a 2D, subsonic potential flow model to explore the properties of the fluid operator. The cascade definition was taken from the Tenth Standard Configuration (Fransson and Verdon, 1993), which is a modified NACA 0006

cascade commonly used in computational flutter studies. A wide parameter space was tested, spanning Mach number, flow angle, and reduced frequency. The results confirmed that compressibility effects, even in the subsonic regime, can have a significant effect upon stability (a difference of 30% in critical reduced frequency between Mach numbers 0.4 and 0.7), and a substantial difference between undamped stability and the case of unity g/ρ^* . The actual case, which includes shocks, separated flows, and 3D effects, can similarly be expected to require a description in terms of all four parameters.

4.2.4 Preliminary Analysis of Experimental Data

Despite intensive effort in the area of simulation, experimental data remains the only source of information that faithfully includes all of the relevant effects in real machines. Using a simple model for the unsteady fluid coefficient l_o , a procedure for representing the flutter boundaries with four empirical coefficients for subsonic flutter and three empirical coefficients for shock-driven flutter has been developed. These tools were successfully demonstrated in the analysis of flutter data from Fan-C, a NASA research fan. A novel boundary fitting technique which aims towards separating the stable from the unstable points was developed, which was demonstrably better than the method of least-squares fitting points on the boundary.

4.2.5 Research Aspects to be Addressed

To follow the current research, the next phase which is envisioned lies in integrating the understanding of parameter dependencies gained thus far with further experimental data sufficient to quantify the effects of flutter boundary migration. This quantification is necessary to achieve the goal of constructing a rational flutter clearance methodology.

Furthermore, the current analytical research suggests that there is an opportunity to develop a flutter boundary representation which not only facilitates practical development of engines via a rational methodology for assessment of flutter margin throughout the flight envelope, but also facilitates our understanding of the relationship between flight condition, structural properties, and flutter stability. It is essential, however, to link the analysis with the experimental results to solidify this understanding.

The steps associated with the next phase, then, might be summarized as follows:

- 1) Quantify effects of flight condition on flutter stability for a particular compressor type.
- 2) Develop a methodology, using (1), for flutter clearance that accounts for the effects of flight condition.

4.3 References

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5.0 TASK III: UNSTEADY PHENOMENA IN TURBOMACHINERY ENDWALL FLOWS (E. M. Greitzer, C. S. Tan)

5.1 Introduction

The overall objective of this task was to develop a quantitative understanding of two unsteady flow problems associated with compressor tip clearance, which have been discussed in the community for a number of years without real resolution. The first is the effect of compressor asymmetric tip clearance, which induces unsteady flow phenomena on the scale of the compressor circumference (*i.e.* large compared to blade pitch) and results in changes to the so-called steady-state performance of multistage compressors. The second was the role of, and the dynamic processes associated with, tip leakage flow in axial compressor rotating stall inception. This situation involves flow features that are initially on the scale of a blade passage. However, the resulting instabilities are coupled to the annulus eigenmode structure and thus exhibit length scales much larger than the blade pitch. In the following we describe work accomplished on both of these unsteady flow problems including a discussion of a resolution of a number of technical issues needed to bring an adequate understanding on the role of endwall flow on limiting compressor stability.

5.2 Effects of Asymmetric Tip Clearance on Multistage Compressor Performance

The problem of asymmetric compressor tip clearance (Fig. 5.1) is structural in origin (it arises, for example, from casing deformation), and shows one feature of the strong coupling between aerodynamic and mechanical phenomena that occurs in high performance gas turbine engines. Interest in this situation arises because of the resulting impact on compressor stability (stable flow range) and performance (pressure rise and efficiency). In this context, a specific operational question is whether one can regard the loss in performance as an effect of average clearance, or whether one needs to consider the behavior in more depth.

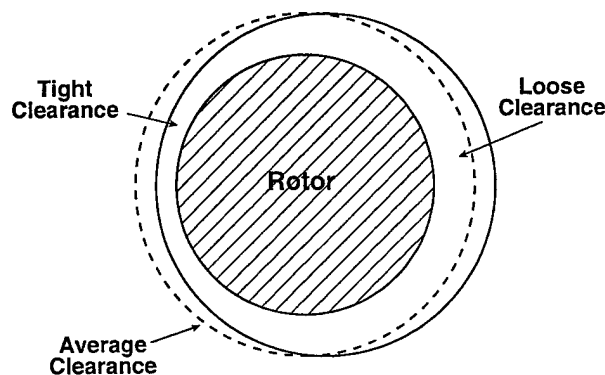


Fig. 5.1: Non-axisymmetric tip clearance caused by off-centered rotor.

The goal of this project (carried out in collaboration with Dr. D.C. Wisler's group at GE Aircraft Engines) was to extract from this complex aeromechanical system a low order description that would appropriately capture the physical essence of the problem. The modeling that was inherent in the research meant that the approach needed to include experiments, as well as analysis, so that the theoretical description could be assessed. The development of the description and the substantiating experiments have both been accomplished in the past three years. A detailed description of this work is given by Graf *et al.* (1998), but we summarize the solution of the problem and the main results herein.

The theoretical model developed utilizes a two-dimensional (x, θ) unsteady description of the flow fields within a multistage compressor and upstream and downstream. The non-uniformities of interest have length scales which are large compared to the blade pitch, allowing a simple description of the behavior of interest, in which the blade passages are modeled essentially as channels with the local unsteadiness accounted for with one-dimensional descriptions of the inertial forces and the transient loss evolution. Clearance asymmetry was analyzed by viewing each circumferential location as operating along a pressure rise characteristic that corresponded to the local level of clearance. To the author's knowledge, the model provided the first nonlinear treatment of steady flow with clearance asymmetry and the first analysis of the effect on compressor stability.

The objective of the model was to provide information about the overall behavior of the compressor and compression system, particularly changes in stability and peak efficiency, rather than to capture detailed fluid mechanics of the clearance flow. Thus, during the course of its development, an issue continually addressed was the degree of sophistication needed to capture the important features of the flow. The results reported in Graf *et al.* (1998) showed that the description adopted has the desired capability.

To assess the utility of the model, two sets of experiments were carried out using the four-stage Large Scale Research Compressor in the Aerodynamics Research Laboratory at General Electric Aircraft Engine Group. A description of this facility is given by Wisler (1985). The low speed research compressor blading used in the experiments was representative of modern design.

In the first set of experiments, compressor characteristics were obtained at different axisymmetric clearance levels; these served as input to the theoretical model. Three clearances were tested corresponding to 1.5%, 2.9%, and 4.3% of annulus height, with the 2.9% clearance configuration serving as the baseline (or nominal) case. For reference, typical levels of clearance-to-annulus height in the rear block of compressors in modern engines are roughly 1.5 to 2.0 percent. Average levels of clearance for the whole compressor are often quoted at about one percent; however, the rear block levels are larger than this average because annulus heights

are smaller. The low speed test geometry for the experiments was meant to simulate the middle and rear blocks of HP compressors, and clearance variations were used that bounded typical rear block engine levels.

The second set of experiments involved two different asymmetric clearance distributions. Theory predicted that the circumferential harmonic content of the clearance variation (and hence of the flow field non-uniformity) would have a significant influence on the severity of asymmetric clearance effects. The test series thus included configurations with a single lobe (first-harmonic) and a two-lobed (second-harmonic) clearance variation.

Both steady-state aerodynamic data and high response measurements were made using circumferential arrays of casing static taps located at several flow measurement planes, mid-span Kiel probes, and casing pressure transducers. Overall efficiency was obtained from shaft speed and torque measurements. Further details of the experimental configuration and instrumentation utilized in this study can be found in Graf *et al.* (1998).

The major findings from this work were:

- 1) The model accurately describes the steady and unsteady behavior of the measured flow field. The effects of clearance asymmetry and of reduced frequency were captured in terms of the effects on pressure rise, flow coefficient at stall, and efficiency.
- 2) Non-axisymmetric tip clearance resulting from casing distortion can substantially reduce the compressor peak pressure rise and stable flow range relative to values obtained for an equivalent average axisymmetric clearance. This is shown in Fig. 5.2 in which the compressor pressure rise characteristics with non-axisymmetric clearance are shown as data points while the axisymmetric ones are indicated by the dashed lines. The 2.9% clearance axisymmetric data can be compared with the non-axisymmetric data, all of which are for an average clearance of 2.9%. As shown by both analysis and experiment, the decrease in stability (stall margin) for an asymmetric clearance was closer to that obtained with axisymmetric flow at the maximum clearance than to that at the average clearance. Thus, the degradation in peak pressure and stable flow range is more nearly associated with the *maximum* level of the clearance asymmetry, rather than the average.
- 3) The reduction in *peak* efficiency due to clearance asymmetry was roughly equal to that obtained for an equivalent average axisymmetric clearance. Thus, peak efficiency degrades in proportion to the increase in average level of the clearance asymmetry.
- 4) The spatial harmonic content of the clearance asymmetry is an important factor in determining the severity of the impact on compressor performance and stability. Asymmetries with lower harmonics are more detrimental than those with higher harmonics, and the largest effects are produced by clearance asymmetries with dominant wavelength

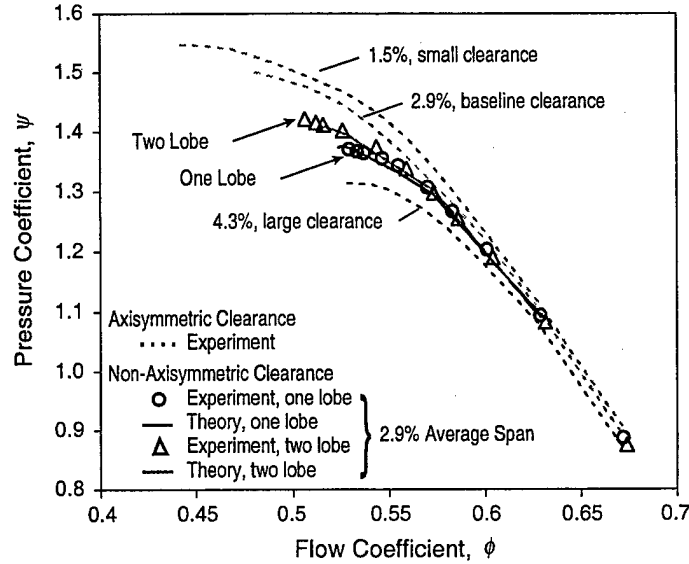


Fig. 5.2: Measured and predicted pressure rise characteristics with non-axisymmetric clearance compared with that for axisymmetric clearance.

equal to that of the compressor circumference. This effect was also captured by the theory (see Fig. 5.2). Another view of this is shown in Fig. 5.3 which portrays the axial velocity variation around the compressor for the single-lobed and two-lobed clearances at different flow coefficient. As calculated, the amplitude of the non-uniformity increases as the flow coefficient decreases. In addition, the amplitude is less for the two-lobed pattern than for the

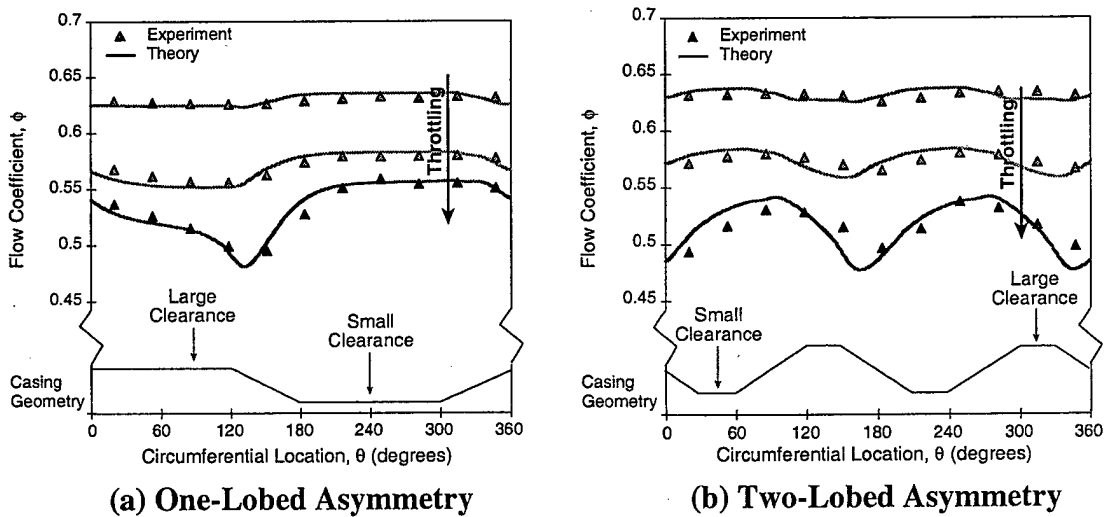


Fig. 5.3: Circumferential distribution of flow coefficient for one-lobed and two-lobed clearance asymmetry.

single-lobed, due to the increased effect of unsteadiness with the latter. This causes (again as described by the model) a smaller loss in stall margin.

- 5) Unsteady flow measurements with asymmetric clearance showed that the variation in RMS unsteadiness around the annulus is related to stall cell inception location and to the disturbance wave structure. The measured location of maximum flow field unsteadiness corresponded to the end of the large clearance sector (as seen by the rotor) which was the theoretical location of maximum disturbance growth. The results imply that the stability of a compressor with clearance asymmetry can be linked to the unsteady flow processes associated with the non-uniform flow field.
- 6) A parametric study was conducted to explore the effects of asymmetric clearance on compressor performance and stability. The results of the study were:
 - There is an increased sensitivity to clearance asymmetry for compressors with characteristics that are steep, have high peak pressure rise, narrow map width, and sharp drop in pressure rise after the peak.
 - Decreasing the wavelength of the clearance non-uniformity (*i.e.* increasing the reduced frequency) decreases the effect of clearance asymmetry on stall margin. There is virtually no penalty when the asymmetry has four or more lobes.

The results from this work have been presented at the 42nd Inter-national Gas Turbine and Aeroengine Congress and Exposition, Orlando USA, June 1997 and published in ASME Journal of Turbomachinery (October 1998 issue).

5.3 Role of Blade Passage Flow Structures in Axial Compressor Rotating Stall Inception

The second topic concerned the role of blade passage flow structure (*i.e.* tip clearance flow) on stall inception. The marked effect of axial compressor tip clearance on stable flow range is a trend which is well documented from an overall performance point of view (Smith, 1970; Koch, 1981; Cumpsty, 1989). In terms of knowledge of the basic mechanisms, however, the phenomenological links connecting tip clearance flow features to the onset of rotating stall (the event that sets the limit on the stable flow range) have not been well identified. In particular, on a blade passage scale, there is no accepted qualitative, let alone quantitative, description of the dynamic processes associated with transition from a situation in which the flow has blade-to-blade periodicity (including the embedded tip clearance vortex) to the strong asymmetry which characterizes rotating stall. This effort was thus focused on addressing the connection between tip clearance flow phenomena and rotating stall inception. The results illustrate the role played by the clearance vortex structure in one route to compressor instability.

The work was motivated by the observation that two types of rotating stall inception occur in axial compressors. The first, which is characterized by waves with length-scales on the

order of the circumference of the compressor, and propagation speed of one-fourth to one-half of rotor rotation, has been referred to as modal stall inception. A number of investigations have been conducted of this phenomenon (*e.g.* Haynes, Hendricks, and Epstein, 1994; Tryfonidis *et al.*, 1995), and a main conclusion is that modal development of rotating stall is associated with the growth of small amplitude sinusoidal (in the circumferential coordinate) flow disturbances. Relatively simple models which view the blade passage as a one-dimensional channel (Moore and Greitzer, 1986; Haynes *et al.*, 1994) yield good predictions of the rotational speed, growth rate, and waveform shape of such disturbances. The agreement between experiment and model implies that knowledge of the overall blade row loss (or equivalently pressure rise) and turning characteristics is sufficient to allow useful analysis of the unsteady features of the instability process. In this framework (which is similar to that described in the previous section), the clearance flow structure is only important insofar as it affects the overall loss and turning, and it does not appear to be necessary to describe the passage flow on more than this global level to provide a physically meaningful description of instability inception.

There is, however, another very different route to rotating stall, characterized by the appearance of disturbances with a dominant length-scale much shorter than the circumference, typically on the order of several blade pitches. These disturbances have a propagation speed roughly 70-80% of rotor frequency (Day, 1993; Camp and Day, 1997) which is higher than that associated with modal disturbances. There is currently no mechanistic description of this phenomenon, which has been referred to as short length-scale stall inception or "spikes". Experiments show that this type of rotating stall inception possesses a radial structure (Silkowski, 1995) and that changes in the size of the tip clearance as well as the first rotor incidence (Camp and Day, 1997) can modify the type of stall inception observed (Day, 1993).

The observations made imply that a different, and more detailed, approach is essential to capture the development of such short length-scale stall inception. More specifically, it appeared necessary to include a description of the clearance flow structure within the individual blade passages. In this connection, in view of the pronounced effects which changes in tip clearance have on compressor stability, an important goal was to clarify the link between tip clearance flow and rotating stall onset.

A specific fluid dynamic question addressed, therefore, is what is the role of the tip clearance flow, and the tip clearance vortex, in the stall inception process. The physical features that need to be included in grappling with this question are as follows:

- i) There are multiple blade passages involved in a rotating stall cell. A potentially larger range of length-scales must be able to be captured by the numerical method than for the computation of flow in periodic blade passages.
- ii) The phenomenon is wave-like and has a larger range of length- and time-scales than the

modal disturbances. This wave behavior needs to be appropriately resolved.

- iii) The compressor operating regime is off-design and the viscous effects which give rise to separated flow must be included. A related consideration is thus the division of the flow into regions of viscous and inviscid flow, as well as the amount of resolution needed for the boundary layers. In the treatment adopted, only flow regions adjacent to the blade surfaces have been treated as viscous so that there are no viscous layers on the casing and hub.
- iv) The three-dimensional flow structure associated with the tip clearance should be explicitly included.

The details of the numerical method that captures all these physical features, and its validation, are described in Hoying *et al.* (1999).

The computational investigation used the geometry of the low speed version of the E³ compressor (Wisler, 1985), which had been found to exhibit a short length-scale type of stall inception (Silkowski, 1995; Park, 1994). This compressor has a set of inlet guide vanes followed by four geometrically identical stages with a hub-to-tip radius ratio of 0.85. The rotors have a blade aspect ratio of 1.2 and a maximum tip Mach number of 0.2. In the experiments cited, stall inception location was in the first stage rotor with the strongest non-uniformities occurring near the tip.

Based on the results of Camp and Day (1997) as well as on the experiments which showed the initial development of rotating stall was confined to the first rotor, a single blade row geometry could be used for analysis of the initial phase of the rotating stall development. The computational domain is a portion of the first rotor consisting of eight blades out of the 54 blades present. The number of eight compressor blades was selected based on (two-dimensional) numerical experiments with the same geometry, which showed that eight blades were capable of also exhibiting a modal stall inception (Hoying, 1996). It is important to have the capability for exhibiting both types of stall inception so as not to prejudice the results.

Clearances of 1.3% and 3.0% (of chord) corresponding to those examined experimentally were numerically simulated. In the computations, as in the experiment, the results were qualitatively similar between the small and large clearances and only the results from the 3.0% case will be discussed.

The steps in the overall computational procedure involved computing the "axisymmetric" performance using a single periodic blade passage calculation. The single passage computation was performed to define the flow regimes of interest as well as to permit a comparison between axisymmetric and non-axisymmetric flow fields. Once the flow regimes had been delineated, a multiple blade passage computation was carried out with a throttle-induced transient to allow for the presence of asymmetric flow. The details of the stalling process were then analyzed much as previous experimental information had been.

Figure 5.4 shows the rotor pressure rise versus flow coefficient for both the steady-state single blade configuration and the transient eight-blade computation (following a single blade passage). The numbered points shown correspond to axisymmetric flow behavior at and past stall, as obtained from the single blade passage calculation. The substantial difference in the behavior near point 1 represents the quantitatively different flow fields between the blade-to-blade periodic solution and the eight-blade calculation. Prior to stall, the difference lies in the unsteadiness in the multi-blade passage calculation, where pressure fluctuations are accentuated by the axially longer flow domain (higher fluid inertia) required in the computation. The points on the multi-blade passage speedline represent snapshots in time, and thus include both high and low points of the fluctuations, *i.e.* points above and below the mean speedline from the single-blade passage calculation. The stall point, which is characterized by a discontinuity in the slope of the single blade passage speedline and by a temporal fluctuation in pressure rise in the multi-blade computation, is the same as in both calculations. However, flow behavior at and after stall is substantially different as reported in Hoying *et al.* (1999).

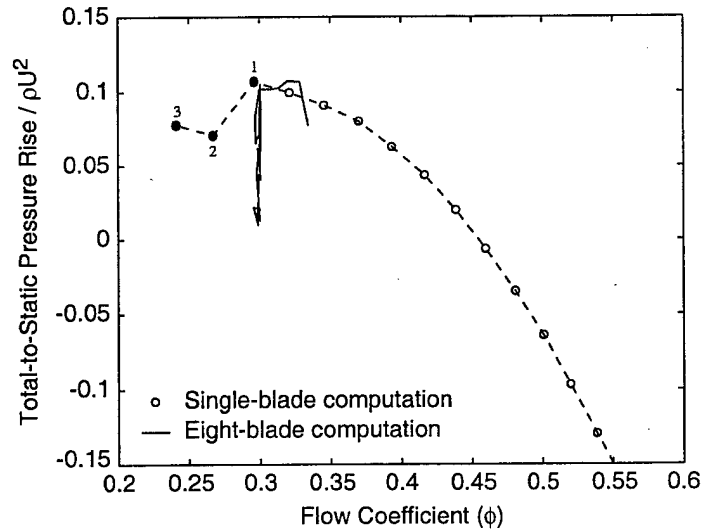


Fig. 5.4: Pressure rise characteristic for the E^3 rotor. Point 1 is the stall point in the single blade passage calculation. Points 2 and 3 are the corresponding results with throttle closed beyond stall. The speedline for the eight-blade calculation shows temporal fluctuations in pressure rise at stall.

The onset of stall could also be linked to the trajectory of the tip clearance vortex. As the flow coefficient was reduced, this vortex moved forward in the passage as shown in Figs. 5.5 and 5.6. Figure 5.5 shows the computed vorticity contour at a point 8% of the span from the tip. Figure 5.6 shows the tip clearance vortex trajectories for different flow coefficients.

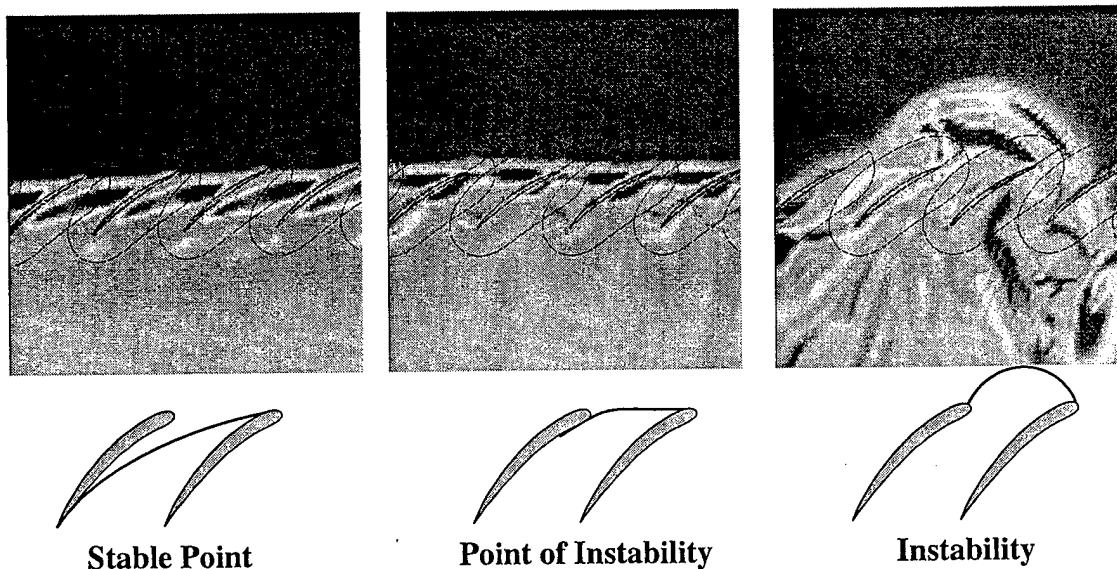


Fig. 5.5: Vorticity contour at 8% immersion of the E^3 compressor indicating the trajectory of the tip vortex before stall, at stall inception, and after stalling has taken place.

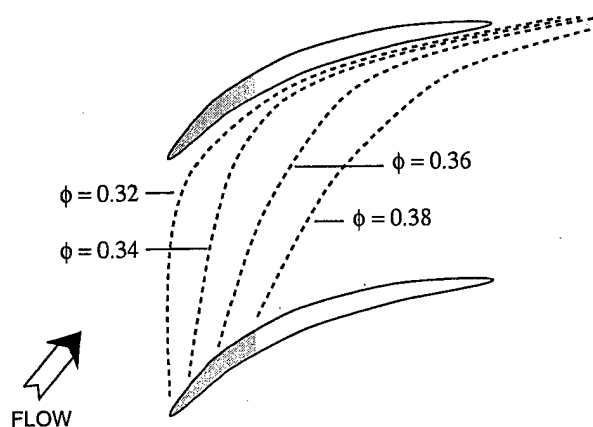


Fig. 5.6: Trajectories of the tip clearance vortex for the E^3 rotor for different flow coefficients.

The limiting stable configuration occurs when the vortex is aligned with the leading edge plane of the blade row. In this situation, as shown by vortex kinematic arguments (Hoying *et al.*, 1999), the vortex is unstable to small disturbances and will move out ahead of the blade passage. The time-dependent computations given in Fig. 5.5 show that this motion leads to a large amplitude axial velocity disturbance which grows in circumferential extent. The computations and the vortex arguments thus suggest that one source of the spikes is the motion of the tip

vortex. Further details can be found in the reference.

Under the (boundary) conditions (isolated rotor row with inviscid compressor casing and uniform upstream flow) for which the computational experiments were implemented, the key results of this part of the research are as follows:

- 1) Computational experiments have been carried out to simulate the short length-scale rotating stall inceptions which are fundamentally different than long length-scale modal stall onset. For the former, inception depends essentially upon local flow conditions, whereas for modal stall inception there is a strong interaction with the rest of the compression system.
- 2) The short length-scale inception process is linked to the behavior of the blade passage flow field structure, specifically the tip clearance vortex. This is in contrast to modal stall, where a description of the flow structures within the blade passages is not required for a useful description of rotating stall inception and development.
- 3) A local stall inception criteria for the short length-scale phenomena, namely the tip vortex trajectory becoming aligned with the blade row leading edge plane, has been identified for axial compressors. This suggests that for tip-critical compressors (as is the case here) single blade passage calculations, rather than computations of the entire annulus, may be able to be used to predict the conditions at which a short length-scale disturbance can occur.
- 4) The origin of this stalling process can also be described in terms of the kinematics of the tip clearance vortex (see Figs. 5.5 and 5.6). Stall inception is a result of the motion of the tip clearance vortex moving out (upstream) of the blade passage, and this occurs when the vortex trajectory is aligned with the blade leading edge plane. The time evolution of the rotor exit blockage is a consequence of this motion.
- 5) The process by which the short length-scale disturbances develop seems generic for axial compressors with tip-critical flow fields so a similar breakdown in axisymmetric flow field should be expected in any compressor which experiences a spilling forward of the tip clearance vortex.
- 6) The present set of calculations is for an isolated rotor and there is a need to examine the instability behavior of the compressor rotor in a multi-blade row environment.

The above results have been presented in the 43rd International Gas Turbine and Aeroengine Congress and Exposition, Stockholm, June 1998 and published in ASME Journal of Turbomachinery (October 1999 issue).

As mentioned above and reported in Hoying *et al.* (1999) the unsteady 3-D multi-blade passage calculations were implemented for an isolated rotor with boundary layers on the blade surfaces but in the absence of casing boundary layers. The results of these calculations lead us to a hypothesis concerning the mechanism behind the origin of the short wave length disturbance route to stall. However, they do not close the issue and the following technical questions need to

be answered: (i) what role does the casing boundary layer play (and how does it affect the behavior of the tip leakage flow), and (ii) what would the behavior be in a multi-blade-row environment?

To begin addressing these questions, several computational experiments have been implemented. Multi-blade passage calculations have been carried out on flow in a rotor stage with no casing boundary layer and with an imposed velocity shear at the casing to simulate the presence of the casing boundary layer.

All of these computations have been post-processed in the same manner as the original computations. The results show that the behavior of the tip leakage vortex is similar to that described in the previous section, namely a movement forward in the passage as the flow coefficient is reduced until the vortex is aligned with the rotor leading edge plane and then instability. These lend support to the hypothesis, but there is a need to carry out the computations including a viscous boundary layer (*i.e.*, one computed based on a non-slip boundary condition at the casing). We thus see the next steps in establishing the validity of the hypothesis to be (i) assessing what aspects of endwall flow structure (tip leakage vortex *vs.* retarded endwall flow due to viscous casing boundary layer) are most critical in determining compressor stall inception in a multi-blade row environment, and then (ii) delineating the causal relationships between specific geometric and aerodynamic compressor characteristics and this fluid dynamic situation. The objectives of this phase of the research are to enable a definitive statement to be made about the validity of the hypothesis, to be able to state what the sequence of fluid dynamic processes is that takes the machine from nominally axisymmetric (*i.e.* variations on the blade passage scale) flow into rotating stall, and to link these to design parameters.

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6.0 PROGRAM PERSONNEL

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7.0 PUBLICATIONS

Graf, M., Wong, T., Greitzer, E.M., Tan, C.S., Marble, F.E., Shin, H-W., Wisler, D., 1998, "Effects of Asymmetric Tip Clearance on Compressor Stability," *ASME Journal of Turbomachinery*, Vol. 120, No. 4.

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8.0 NEW DISCOVERIES, INVENTIONS, OR PATENTS

“Counter-Rotating Compressors with Control of Boundary Layer by Fluid Removal”, A.H. Epstein and J.L. Kerrebrock, submitted Jan. 1997.

“Enhancement of Turbomachines and Compressors by Fluid Removal,” J.L. Kerrebrock, submitted Jan. 1997.

9.0 INTERACTIONS

There are considerable interactions between the Gas Turbine Laboratory personnel and industry and government, as well as with other universities. This is accomplished through collaborative research projects with industry (Pratt & Whitney, AlliedSignal, ABB, General Electric Aircraft Engines and Power Systems, and Solar Turbines) and government laboratories (NASA Glenn and WPAFL) and an active seminar program that brings speakers from industry and government. The speakers during the AFOSR grant period include:

Dr. John Adamczyk, NASA Lewis Research Center
"Modeling of Multi-Stage Turbomachinery Flows"

Mr. Scott Copeland, United Technology Research Center
"Active Control in Compressor Flutter"

Prof. Nick Cumpsty, Whittle Laboratory, Cambridge University
"Real 3D Flow in Turbomachines"

Mr. Otha Davenport, Wright-Patterson AFB
"USAF Gas Turbine Trends Into the Next Century"

Dr. Mark Glauser, AFOSR/NA, Clarkson University
"Multi-Point Measurement and Low-Dimensional Models: Tools for the Characterization and Control of Turbulent Flows"

Dr. Richard Murray, United Technologies Research Center
"Nonlinear Dynamics and Control of Fluids with Applications to Gas Turbine Engines"

Dr. Alan Pfeffer, ABB
"Applications of Fluid Mechanics in Circulating Fluidized Beds"

Dr. Richard Price, Pratt & Whitney Aircraft
"Turbine Design - Yesterday, Today and Tomorrow"

Mr. Jayant Sabnis, Pratt & Whitney Aircraft
"Gas Turbine Engine Internal Air Systems"

Mr. Alan Weaver, Pratt & Whitney Aircraft
"Achieving Safety with Data Analysis"

Dr. David Wisler, GE Aircraft Engines
"Engineering - What They Don't Necessarily Teach You in School"